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Numerical Simulation of the Oscillations in a Mixer—An Internal Aeroacoustic Feedback System

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Numerical Simulation of the Oscillations in a Mixer—An Internal Aerocoustic Feedback System

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Summary. The space-time conservation element and solution element method is employed to numerically study the acoustic feedback system in a high temperature, high speed wind tunnel mixer. The computation captures the self-sustained feedback loop between reflecting Mach waves and the shear layer. This feedback loop results in violent instabilities that are suspected of causing damage to some tunnel components. The computed frequency is in good agreement with the available experimental data. The physical phenomena are explained based on the numerical results.

1 Introduction

In modern hypersonic test facilities, mixers [1] can be used to allow a wind tunnel to operate at different Mach numbers thus making the test facility more versatile. An example of such a tunnel is the NASA Langley eight foot high temperature tunnel. The original axisymmetric nozzle was designed to provide a Mach 7 flow condition at the test section with a diameter of eight feet. A tunnel modification shortens the nozzle, adds a mixer, and an additional nozzle to provide the Mach 4 flow condition at the second nozzle exit where it enters the test section. The mixer is used to combine the high pressure, and high temperature jet core with the ambient air to generate an appropriate flight environment for wind tunnel testing. The mixer must be carefully designed to avoid any unwanted performance degradation caused by unsteady flow phenomena such as high amplitude oscillation. Such oscillations and consequential waves could potentially damage or destroy the tunnel as already observed in the NASA Langely mixer. Figure 1 illustrates the streamwise cross section of the wind tunnel upstream of the experimental test section where the flow exhibits such unwanted unsteady characteristics. This section of the wind tunnel consists of two converging-diverging (C-D) nozzles serially connected by a large mixing chamber. The inlet (upstream condition for the first C-D nozzle) is supplied by a high temperature and high pressure combustor. In the diverging section of the first C-D nozzle, the flow expands and then separates at the expansion lip as it enters the mixing chamber.

Vortices or shocklets are shed and grow in strength along the shear layer. Consequently, strong Mach radiation waves are generated in a feedback cycle in which the reflected Mach waves from the mixing chamber wall propagate upstream around the jet core to the exit of the first C-D nozzle to excite new vortices or shocklets. The phenomenon is complicated and interesting, and to our best knowledge, has not yet been studied numerically.

Numerical investigation of such a problem with mixed aeroacoustic waves and shocks poses stringent requirements that the scheme must be able to: (i) resolve acoustic waves with low dispersion and dissipation, (ii) capture shock waves, and (iii) have an effective non-reflecting boundary condition. The space-time conservation element and solution element method (CE/SE), is a numerical method that meets the above requirements and is adopted in this paper.

Background information on the governing equations and computational grid are provided first followed by the problem description, results, and conclusions in Sections 2 through 5 respectively.

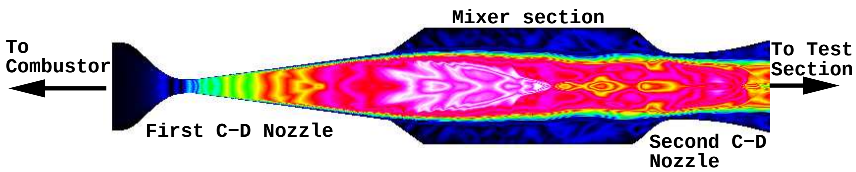

Fig. 1. Computational domain and instantaneous isomach contours, Mach number range 0.0 to 5.5.

2 Governing Equations, Numerical Scheme, and Grid

The unsteady axisymmetric Navier-Stokes equations are used to model the viscous problem in this study. As explained in Section 1, the nonlinear aeroacoustic feedback loop lies primarily in the streamwise direction. The two-dimensional axisymmetric Navier-Stokes equations are chosen as the governing equations for computational efficiency. Therefore, some possible three-dimensional flow phenomena of secondary importance such as the helical mode, etc. may be excluded. In nondimensionalized differential form the conservation laws are written as:

$$\mathbf{U_t} + \mathbf{F_x} + \mathbf{G_y} = \mathbf{Q}, \tag{1}$$

where x, y, and t are the streamwise and radial coordinates and time, respectively. The conservative flow variable vector is $\mathbf{U}$, and the flux vectors are $\mathbf{F}$ in the streamwise direction and $\mathbf{G}$ in radial direction. The flux vectors are a composite of the inviscid and viscous fluxes. The source term, $\mathbf{Q}$, appearing on the right hand side of the equation, Eq. (1), is a result of the axisymmetric formulation [3].

Considering (x, y, t) as coordinates of a three-dimensional Euclidean space, E_3, the Gauss divergence theorem can be used to give the following equivalent integral form of the conservation laws, Eq. (1):

$$\oint_{S(V)} \mathbf{H_m} \cdot \mathbf{dS} = \int_{\mathbf{V}} \mathbf{Q_m dV}, \qquad \mathbf{m = 1, 2, 3, 4}, \tag{2}$$

where $S(V)$ denotes the surface of the control volume V in E_3 and $\mathbf{H_m} = (\mathbf{F_m}, \mathbf{G_m}, \mathbf{U_m})$ represents the total flux leaving the boundary of space-time region. The integral form of the conservation laws allows the capture of shocks and other flow discontinuities naturally. This integral form of the equations is solved numerically by a recently developed finite volume method, the space-time conservation element and solution element (CE/SE) method. Details of the CE/SE method can be found in [4, 5]

The CE/SE method is constructed to take advantage of unstructured grids for complex geometries. In this case the geometry is relatively simple, so a structured (quadrilateral) element based grid is first generated. The cells are then subdivided into four separate triangular elements and converted to an unstructured format compatible with the flow code. The computation is axisymmetric, so only the upper half of the domain needs to be considered.

For aeroacoustic problems, large numbers of cells are typically needed to resolve the unsteady waves. In this case a subdivided $50X655$ point mesh results in a 128184 triangular cell grid. The grids are made more manageable through the use of parallel computer architecture. The unstructured mesh is decomposed into eight subdomains to take advantage of computers with multiple processors. A message passing library is used for point-to-point communication between processors.

3 Problem Description

The physical domain of interest for the problem is shown in Fig. 1. The inlet conditions for this problem are specified from the exit of the combustor. The high pressure high temperature combustor flow enters the nozzle from the left, chokes in the nozzle throat, expands, and enters the mixer. The flow then passes through the second nozzle and expands into the downstream test section.

We are interested in predicting unsteady flow field upstream of the test section. The diameter, D, at the throat of the upstream nozzle is chosen as the length scale. The density, speed of sound, and temperature, at ambient conditions are taken as scales for the dependent variables.

Initially, the flow in the computation domain is set to ambient flow conditions, *i.e.*, (using nondimensional variables) $\rho_a = 1, \quad p_a = \frac{1}{\gamma}, \quad u_a = 0, \quad v_a = 0.$

At the inlet boundary the conservative flow variables and their spatial derivatives are specified to be those of the combustor exit flow conditions: $\rho_0 = 25.099, \quad p_0 = 124.294, \quad u_0 = 0, \quad v_0 = 0$, with all the derivatives set to zero. The nondimensional flow variables at the exit, downstream of the second C-D nozzle, are specified as ambient.

At the symmetry axis, *i.e.*, $y = 0$, a simple reflective boundary condition is applied. The no-slip boundary condition is applied on the nozzle walls.

4 Results

Figure 1 is a snapshot of the computational domain illustrating the Mach number contours in the domain. The flow enters the first C-D nozzle at a very low Mach number ($M = 0.005$). The flow chokes in the nozzle throat and expands in the diverging section to about $M = 5$ in the core. At the sudden expansion lip, entering the mixing section, the flow separates and shock cells form. The core does not fill the mixer but passes through and enters the second C-D nozzle. The resulting separated annular region surrounding the core remains subsonic allowing waves to propagate upstream. The Mach radiation waves Fig. 2 are formed when the local phase speed of the outer skirt of the unstable shear layer reaches supersonic. This Mach wave strikes the converging section of the second C-D nozzle and the reflected wave propagates upstream. This wave reaches the sudden expansion lip in the mixer section where the receptivity is high triggering another vortex street and the feedback loop is thus completed. The presence of screech in this calculation is indeterminate.

A sequence of snapshots is illustrated in Fig. 3. Here the shock cell deformation is more obvious. Also the effect of the wave propagation in the subsonic region can be observed. The streamwise propagating Mach waves show the interference from the upstream propagating reflected waves. Shocklets/vortices can be seen along the shear layer causing a rippling effect apparent in the shock cell. These vortices or shocklets are shed and grow in strength, propagating downstream to reflect from the converging section of the second C-D nozzle.

A time history of pressure data is recorded, Fig. 4, at several specified points in the computational domain. This data is then post processed by Fast Fourier Transform techniques to obtain spectral information. A large number of times steps are required to achieve appropriate accuracy in the Fourier analysis of the time series data. Small time steps are required initially to keep the code stable. Over three million time steps are needed to remove the initial transients from the computational domain. Another 1.3 million time steps are required to record the time series data for spectral analysis. The computation and spectral analysis are continued until a pervasive fundamental frequency appears. Over seven million time steps are computed for this test case.

Data was recorded at several points in the flow field. The most interesting spectrum was obtained at the point $x/D = 30.0, y/D = 2.5$ which is in the subsonic region of the mixer near the second C-D nozzle. A plot of the analysis is illustrated in Fig. 5. Two very strong peaks at $730Hz$ and $1440Hz$ are indicated. The high frequency is obviously the harmonic of the fundamental frequency due to strong nonlinearity. The reported experimental data [2] for this configuration gave peak values at $800Hz$ and $1400Hz$. The strength of these waves ($\sim 160dB$) is suspected to be strong enough to loosen retaining bolts and destroy the liner of the mixer, as observed in the experiment.

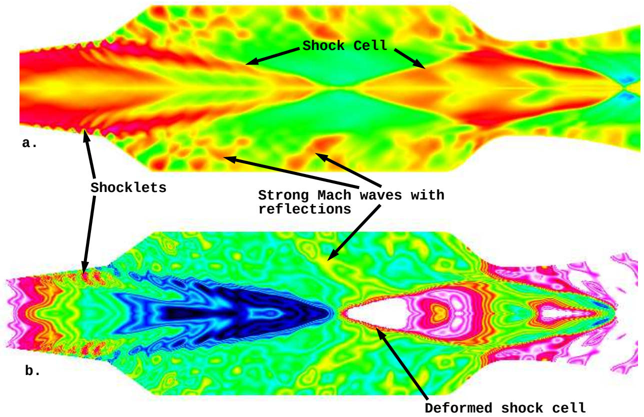

Fig. 2. Snapshot of (a.) radial velocity and (b.) isobar contours showing deformed shock cells, Mach waves, and shocklets/vortices, plotting contours -1.70 to 1.45 and 0.1 to 0.8 respectively.

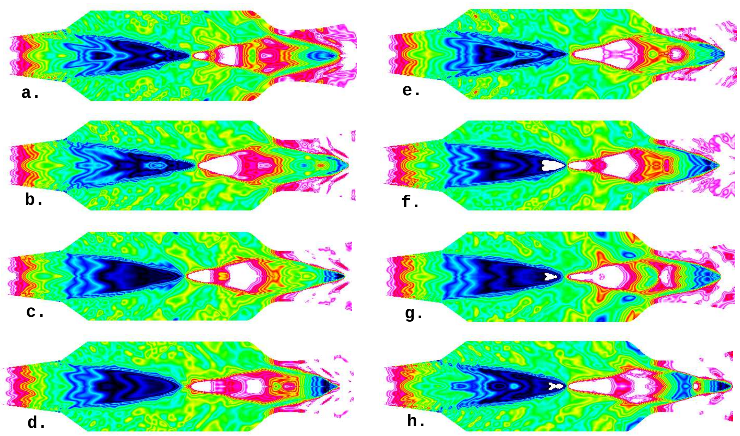

Fig. 3. Sequence of snapshots with isobars contours a-h, pressure contour range 0.1 to 0.8.

5 Concluding Remarks

In this paper, the unstructured space-time CE/SE Navier-Stokes method is applied to a modified wind tunnel mixer to understand the physics of the acoustic feedback system. The results are analyzed for the spectral content. The computed frequency corresponds well with the experimental findings [2] at Mach 4. For further work, it may be necessary to model more of the tunnel operating conditions at a higher fidelity and dimension to truly understand the flow physics of the wave structure in the tunnel in conjunction with physical experiments.

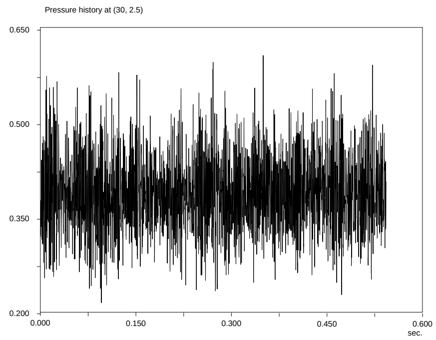

Fig. 4. Time history of pressure at the point $x/D = 30.0, y/D = 2.5$.

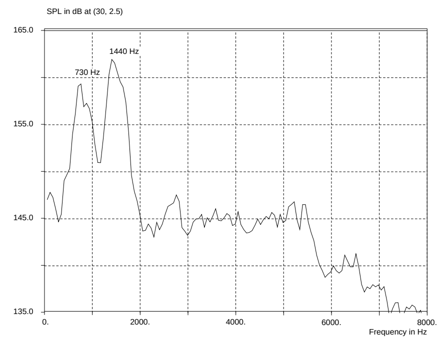

Fig. 5. Spectral analysis at the point $x/D = 30.0, y/D = 2.5$.

A computational tool that can analyze wind tunnels over a wide Mach number range capable of accurately capturing the unsteady wave propagation and interactions could circumvent costly structural damage.

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